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BOUNDARY LAYER TRANSITION

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ABSTRACT

The boundary layer stability, its active control by sound and surface heating and the effect of curvature are studied numerically and experimentally for subsonic flow. In addition, the experimental and flight test data are correlated using the stability theory for supersonic Mach numbers.

Active transition fixing and feedback control of boundary layer by sound interactions are experimentally investigated at low speed over an airfoil. It is shown that a nonintrusive narrow heating strip causes abrupt changes in velocity profile and triggers instant transition at favorable pressure gradients. Sound interaction at normal incident angles produces significant reduction in velocity perturbations in the region of transition.

Numerical simulation of active control by surface heating and cooling in air shows that by appropriate phase adjustment a reduction in the level of perturbation can be obtained. This simulation is based on the solution of two-dimensional compressible Navier-Stokes equations for a flat plate.

Görtler vortices are studied experimentally on an airfoil in the Low Turbulence Pressure Tunnel (LTPT). The flow pattern is visualized using the sublimating chemical technique and data are obtained using a three component laser velocimeter. It is observed that the vortex wavelength is preserved in the streamwise direction but varies with the Görtler number as predicted by linear stability theory.

The effect of curvature on swept leading-edge stability on a cylinder is numerically studied. The results suggest that transition is dominated by traveling disturbance waves and that the wave with the greatest total amplification has an amplitude ratio of e^{11}. Without the curvature this ratio is increased to e^{17}.

Experimental data from the "quiet" supersonic tunnel and flight tests are analyzed using linear compressible stability theory. The data are obtained on a 5-degree half angle cone with unit Reynolds numbers between 9 to 27 million. The analysis shows that transition could be correlated by the e^N method with N in the range of 9 to 11.

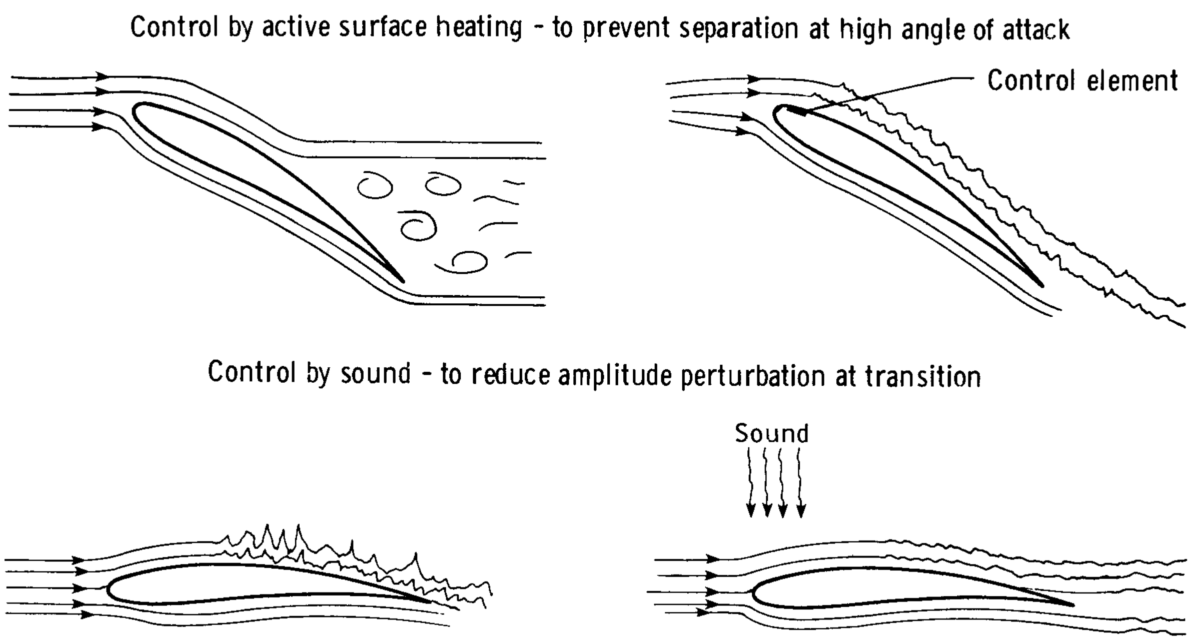
ACTIVE CONTROL OVER AN AIRFOIL

This presentation describes an experiment on active control over an airfoil surface in air. This experiment was conducted at CALTECH where Liepmann et al. (ref. 1) in an earlier experiment demonstrated flow control by active surface heating in water.

Two concepts of control were investigated. The first was active transition fixing by surface heating in the region of favorable pressure gradient to prevent laminar separation at high angles of attack. In this region the flow is highly receptive to surface heating and one can trigger small or large amplitude disturbances as well as trigger instant transition with a single control element shown schematically in the upper part of the figure.

The second control concept investigated was the interaction of sound at near normal incidence with transitional flow over the airfoil to reduce amplitude of the fluctuations.

METHODS OF CONTROL



RESULTS

The photograph (next page) on the top left shows a view of the airfoil with two sets of surface heaters mounted flush with the surface - one each in the favorable (near leading edge) and unfavorable pressure gradient (downstream) regions. The experiment was conducted at freestream velocities up to 12 m/s with corresponding Reynolds number of 3.6×10^6 per meter.

Active transition fixing was accomplished by exciting the flow with a wave packet input to the heaters (HO) located near the leading edge. The output response was recorded by a hot-wire (HW) located downstream. The velocity perturbation as a function of time is shown in the bottom left picture where abrupt changes indicate instant transition. Similar input to the heaters located in the unfavorable pressure gradient region showed only marginal effects. This shows that the flow is receptive to outside disturbances only in the region of favorable pressure gradient.

The figures on the right show the effectiveness of boundary layer control by sound interactions in the region of transition. The sound was produced by a speaker mounted in the wall of the tunnel above the airfoil. The speaker was driven by a feedback loop between the heater input and hot-wire output. The top picture shows the uncontrolled response, while the bottom one shows the feedback controlled response. It is evident that dramatic amplitude reduction is achieved. Similar reduction was noticed with pure tone and random signal. It is observed that at the control output the amplitude of the lower frequencies is reduced drastically at the expense of an increase in the background disturbance which is dominated by higher frequencies. Thus, it is clear that the flow will not return to its uncontrolled state.

In conclusion, these methods are powerful and practical techniques of flow control. Nonintrusive transition fixing could be utilized to prevent separation in ducts as well as augment maximum lift on airfoils. Interaction of sound with the flow is an effective way to control amplitude growth even for transitional flow.

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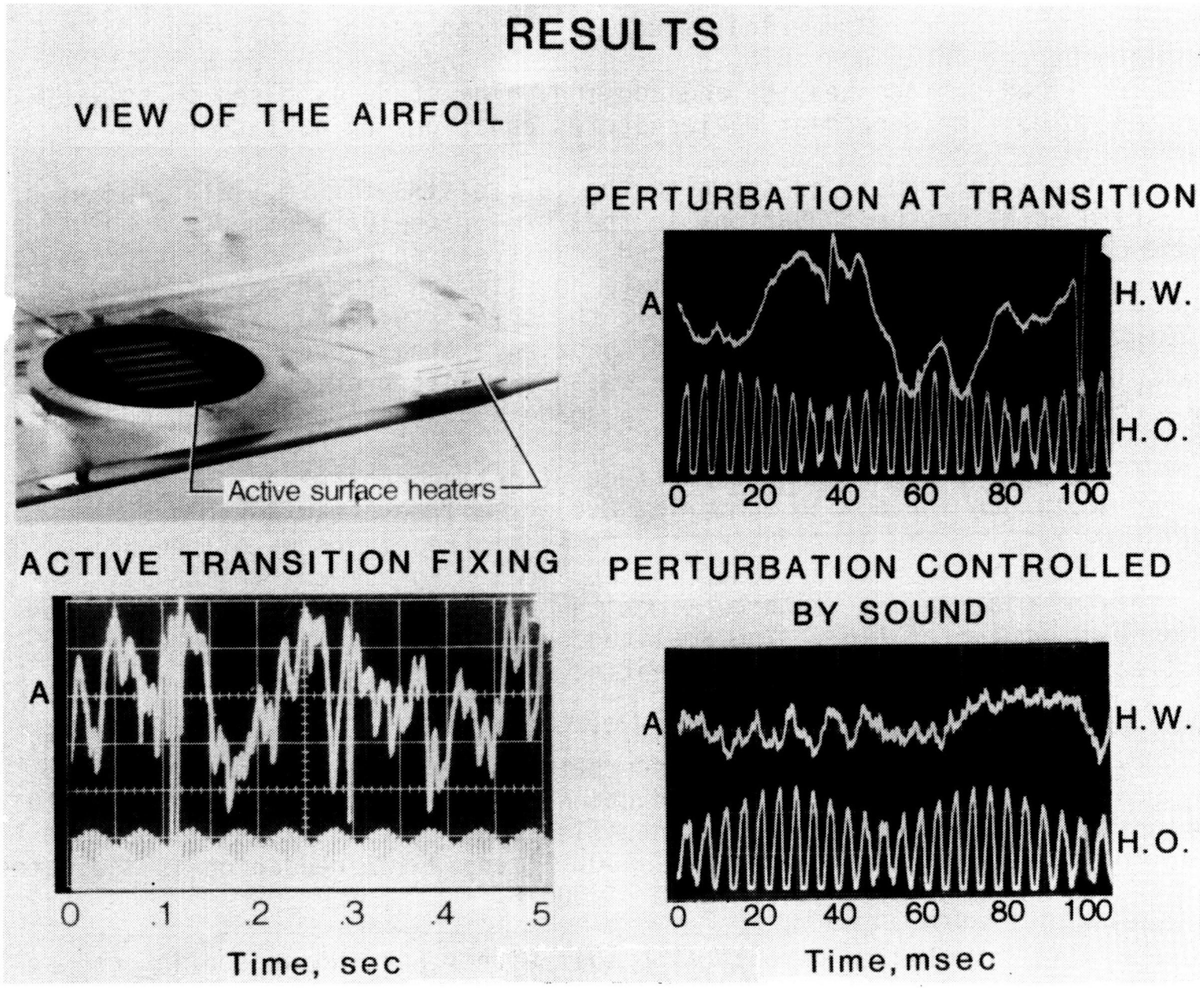


AMPLITUDE CONTROL BY HEATING AND COOLING

This figure concerns a numerical study of the concept of active control by growing disturbances in an unstable, compressible boundary layer by using time periodic, localized surface heating and cooling. The study is based on solving 2-D, compressible, time dependent Navier-Stokes equations on a flat plate.

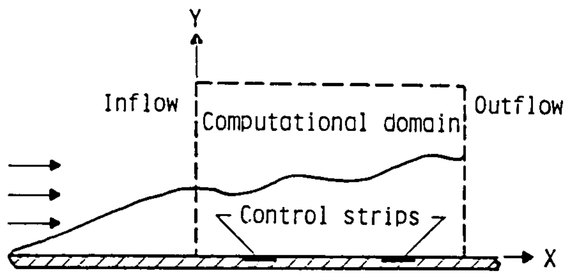
The computational domain is shown on the left of the figure. Starting with a steady state solution, perturbations in the form of the Orr-Sommerfeld solution are superimposed at inflow. At the top and downstream boundaries characteristics radiation conditions are used. At the plate no slip and specified temperature boundary conditions are used. For control, the temperature boundary condition is modified locally over the width of the strip using a steady and unsteady component with phase input. The method of solution is an explicit predictor-corrector technique which is fourth order accurate in space and second order in time.

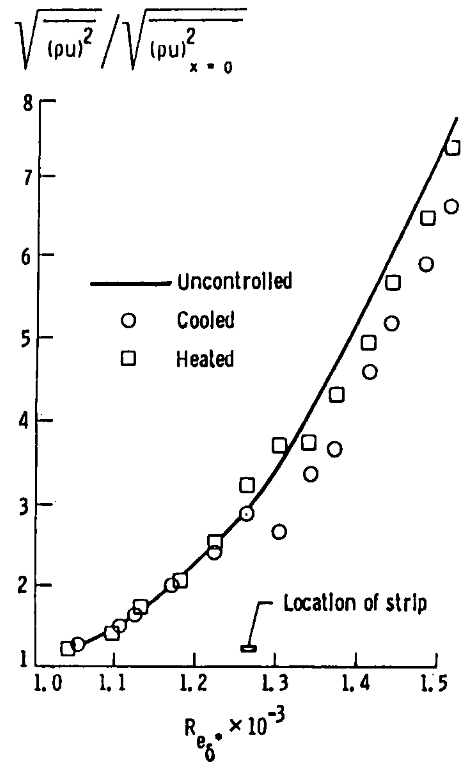
The figure on the right shows the effect of active control on the growth of disturbance amplitude for a Mach number of 0.4 and a freestream Reynolds number/foot of 3×10^5. The disturbance growth is plotted in terms of the RMS of mass flux versus Reynolds number based on the local displacement thickness. The control strip of width equal to three times the displacement thickness is located at the Reynolds number of 1263. The amplitude growths are compared between the uncontrolled and controlled cases with heating and cooling.

Since steady state heating is destabilizing in air while cooling stabilizes, it is necessary to use a 180° phase difference between heating and cooling for unsteady control. The figure shows a reduction in the amplitude growth for both heating and cooling compared to the uncontrolled case. A reduction of about 6% is indicated for heating with an overheat of 1000°F, and a 12% reduction is indicated with cooling for a temperature difference of 300°F.

The numerical simulation demonstrates that either heating or cooling can be used effectively to reduce the level of growing disturbances in a boundary layer. A larger reduction can be obtained by use of multiple control strips placed successively downstream with appropriate phase adjustment.

AMPLITUDE CONTROL BY HEATING AND COOLING





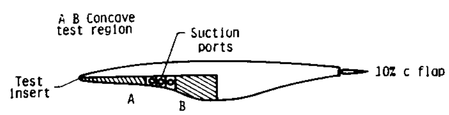
- Solve 2-D unsteady, compressible Navier - Stokes equations
- Control by localized, periodic surface heating and cooling
- $M_\infty = 0.4$, $Re/ft = 3 \times 10^5$

GÖRTLER VORTEX EXPERIMENT

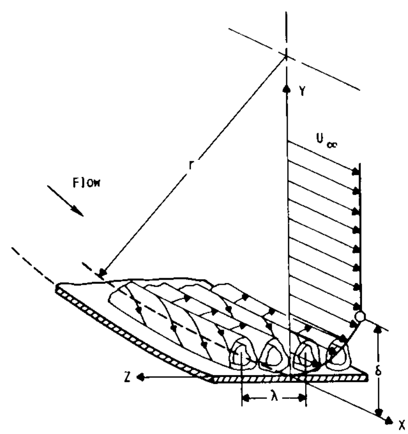
Görtler vortices arise in boundary layers along concave surfaces due to centrifugal effects and are one of three known types of flow instabilities that lead to boundary layer transition. Advanced laminar-flow control (LFC) supercritical airfoils have concave curvature near the leading edge of the lower surface, as shown in the schematic diagram for the airfoil model. Pairs of counter-rotating, streamwise vortices, well known as Görtler vortices, develop in the concave region as shown in the accompanying sketch for the vortex flow pattern. Here, U_∞ is the external flow velocity, δ is the boundary layer thickness parameter, r is the radius of curvature, and λ is the Görtler-vortex wavelength.

The 1.83-meter chord airfoil model shown in the schematic diagram was tested in the NASA Langley Low Turbulence Pressure Tunnel (LTPT). Görtler vortices were observed using a sublimation technique and velocity measurements were made with Laser Velocimetry. The airfoil model has a concave test region extending from 17.5% chord to 27.5% chord. The attached laminar boundary layer was insured by means of suction through a 0.11 × 0.76-meter perforated titanium panel located in the compression part of the concave region. The 10% chord flap at the trailing edge was used to adjust the leading-edge stagnation point. The chord Reynolds number was varied from 1.0 million to 5.9 million, yielding a Görtler number range of 29 to 46.

MODEL SCHEMATIC DIAGRAM



GÖRTLER VORTEX FLOW PATTERN



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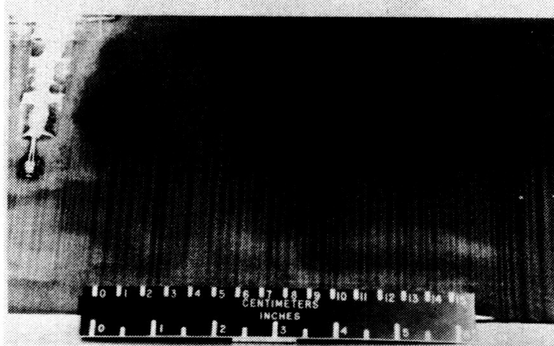
RESULTS

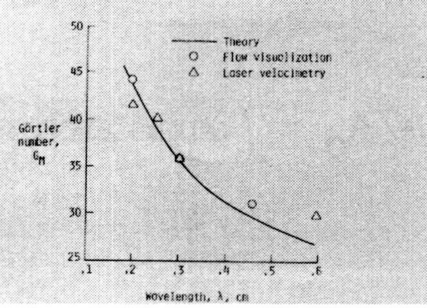
A thin layer of solid white bephenyl material was sprayed on the black model surface to visualize the flow pattern developed due to the presence of Görtler vortices in the boundary layer along the concave test region. The flow pattern is made visible due to the differential surface shear stress distribution under the layer of counter-rotating pairs of streamwise vortices. A set of black and white bands constitutes a pair of vortices and represents the wavelength of these vortices. A representative flow pattern corresponding to a Görtler number of 36 is shown in the accompanying photograph (flow is from bottom to top in the photograph).

Laser velocimeter measurements of streamwise velocities at different chord locations as well as at various heights above the surface were used to determine the vortex wavelength. A fixed, essentially uniform, vortex spacing was observed in the concave zone by both flow visualization and laser velocity measurements at each external flow condition. As in all previous experiments, the dimensional wavelength was preserved in the flow direction, but unlike the earlier experiments, the wavelength was observed to vary appreciably with Görtler number. The variation in the wavelength with Görtler number is shown in the lower figure where it is compared with results based on linear theory obtained by computing wavelength corresponding to maximum amplification conditions.

FLOW VISUALIZATION



VARIATION OF DIMENSIONAL WAVELENGTH
WITH GÖRTLER NUMBER, COMPARISON WITH THEORY

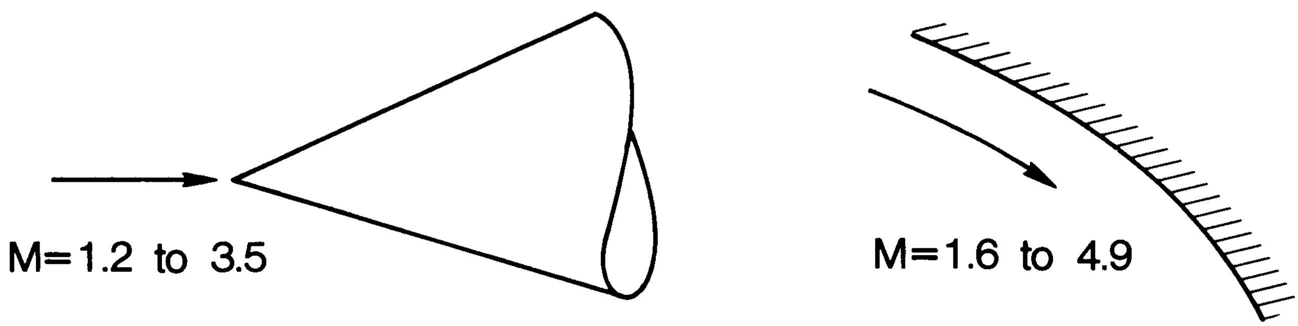


CALIBRATION OF STABILITY THEORY FOR TRANSITION PREDICTION

Linear compressible stability theory was used to analyze supersonic transition data for 10° sharp tip cones at zero angle of attack. Recent flight data at $M = 1.2$ to 1.9 for a cone mounted on the nose of an F-15 aircraft were used. Data obtained in the Langley Mach 3.5 Pilot Low-Disturbance Tunnel were also used. Integration of theoretical disturbance amplification rates for Tollmien-Schlichting (T-S) waves along the cone from the neutral stability point to the measured location of transition onset yields the amplification N factor.

The Görtler instability is usually dominant on concave walls and is observed as small counter-rotating vortices aligned with the flow. Transition was measured for laminar boundary layer flow on the concave walls of quiet wind tunnel nozzles where the local Mach numbers varied from 1.6 to 4.9. The N factors for both types of instabilities varied from about 9 to 11 as illustrated in the next figure.

Mach number > 1



Cone
(T-S instability)
N= 9-11

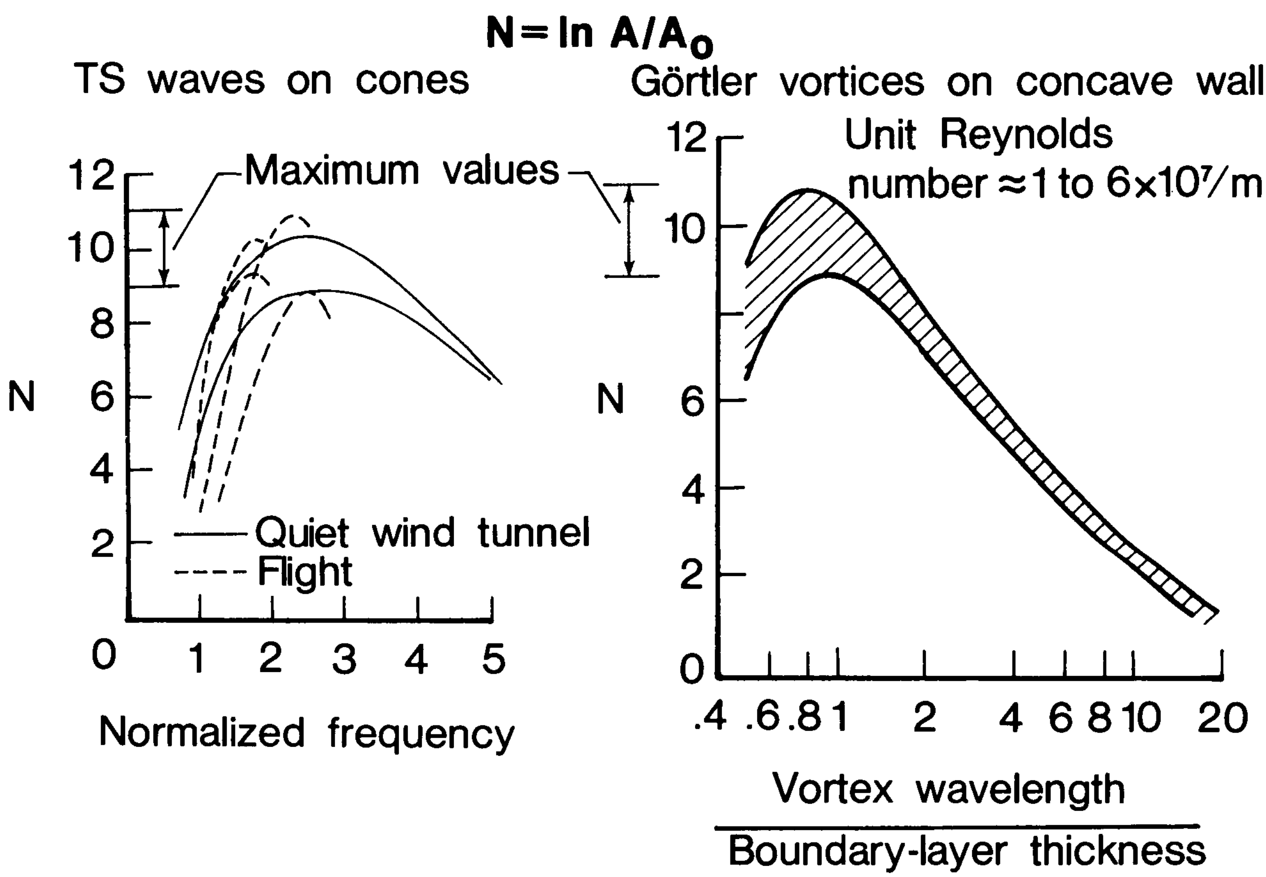
Concave wall
(Görtler instability)
N= 9-11

$$N = \ln A/A_o = \int \text{amplification rate}$$

(REF. 2)

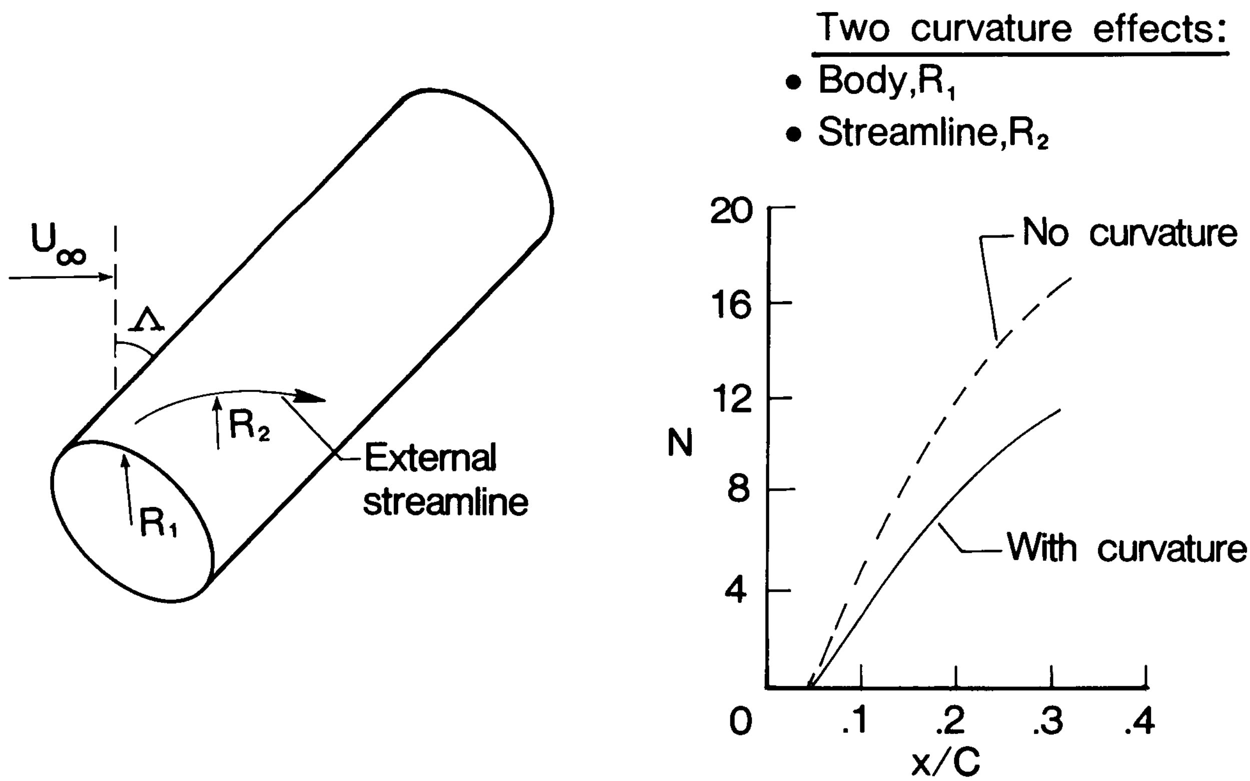
The plot on the left shows the envelope curves N as a function of the dimensionless frequency parameter ($F = 2\pi f\ \nu_e/U_e^2 \times 10^5$) for T-S waves on cones. The maximum amplification occurs at the peak of the curves where N varies from about 9 to 11 at the measured location of transition for both the flight data and the quiet wind tunnel data. This agreement indicates that the low-disturbance environment of flight is correctly simulated in this tunnel.

The plot on the right shows the variation of N with the ratio of vortex wavelength (or vortex width) to boundary layer thickness for the Görtler instability on the concave wall of the Mach 3.5 Pilot Low-Disturbance Wind Tunnel over a range of unit Reynolds numbers. Again, the peak values of N vary from about 9 to 11 at the measured locations of transition. Note that transition always occurred when the vortex wavelength was about the same as the local boundary layer thickness.



CURVATURE EFFECTS ON SWEPT LEADING-EDGE FLOW STABILITY

The stability of the three-dimensional laminar boundary layer on the windward face of a long cylinder was solved by a computational scheme that includes the body curvature R_1 and the streamline curvature R_2. The maximum growth rates are integrated to the measured locations of transition on a cylinder in a low-speed wind tunnel at velocities from 25 to 180 ft/sec. When the curvature terms are omitted, as in previous investigations, the amplitude ratio was approximately e^{17}. When all curvature terms are properly accounted for, the amplitude ratio is about e^{11} in good agreement with the supersonic results of the two preceding figures as well as with various other types of subsonic flows.



(REF. 3)

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